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SPACECRAFT DESIGN SENSITIVITY FOR A DISASTER WARNING SATELLITE SYSTEM

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16. Abstract
A disaster warning satellite system (DWSS) is currently being studied by the National Aeronautics and Space Administration (NASA) at the request of the National Oceanic and Atmospheric Administration (NOAA). Its purpose is to provide the capability to warn the general public of impending natural disasters. A disaster warning satellite (DWS) concept responsive to NOAA requirements and maximizing the use of ATS-6 technology was developed at the NASA Lewis Research Center. Upon completion of concept development, the study was extended to establishing the sensitivity of the DWSS spacecraft power, weight, and cost to variations in both warning and conventional communications functions. The results of this sensitivity analysis are presented in this report.

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SUMMARY

The National Aeronautics and Space Administration (NASA) at the request of the National Oceanic and Atmospheric Administration (NOAA) has studied an advanced disaster warning satellite system (DWSS). The space segment of the DWSS will provide both direct broadcasting (warning) service to the general public and fixed service to the NOAA field organization. A computer technique was developed to establish the sensitivity of spacecraft power, weight, and cost to variations in broadcasting and fixed service capability. The disaster warning satellite (DWS) concept described herein was based on ATS-6 technology to meet technical objectives and to lower total program cost through high inheritance. The study approach and the results of the sensitivity analysis are presented in this report.

INTRODUCTION

Each year natural disasters exact an enormous toll in lives, economic loss, and human suffering in the United States. The loss in lives results, in part, from deficiencies in our warning and preparedness programs. The Disaster Act of 1970 committed the Federal government to assume major responsibilities in disaster preparedness planning and assistance. The Office of Telecommunication Policy (OTP) recommended that the primary responsibility for warning and preparing the general public for any impending disaster be given to the National Oceanic and Atmospheric Administration (NOAA). Since 1970, NASA and NOAA have been jointly assessing the potential of satellite systems to aid in disaster warning. Part of this effort is described in this report.

The definition of the spacecraft communications functions used for the assessment study evolved from a continuous review of NOAA disaster warning satellite (DWS) requirements. The NOAA expressed their initial DWS requirements in a letter sent on

June 17, 1972, to NASA by the associate administrator of NOAA. These requirements are listed in table I.

The Computer Science Corporation (CSC) performed a study (ref. 1) to develop a satellite concept that would fulfill the initial spacecraft communications requirements listed in table I. These requirements resulted in a spacecraft with a weight of 3650 kilograms and a unit cost (recurring cost) of $58 million. Of the communications functions, the requirement for 10 warning channels had a significant impact on spacecraft power requirements and, therefore, on cost and weight (see following section COMMUNICATIONS FUNCTIONS). To examine the warning channel requirement in more detail, a message traffic analysis and simulation was performed at the Lewis Research Center (ref. 2). It was determined that three warning channels would meet the NOAA warning objectives. Following this analysis a "service intent" satellite was defined (table I). The service-intent satellite featured a reduction in the number of warning and mobile reporting channels but an increase in the two-way communication between weather service offices (WSO's)[1] and mobile units. The number of mobile direct channels was doubled, from 5 to 10.

In the summer of 1974, a study was performed at the Lewis Research Center to define development costs for a service-intent spacecraft. The following summer a special study team was formed within the Space Flight Systems Study Office (SFSSO) of Lewis to coordinate the DWS concept development. This team performed studies to determine the effect on a disaster warning spacecraft of varying communications requirements. Upon reviewing the results of these two efforts with NOAA, Lewis was able to establish an appropriate range for the number of each of the needed communications channels to be used in a sensitivity study. This range is presented in table I under "Present spacecraft." The results of the sensitivity study are presented in this report.

COMMUNICATIONS FUNCTIONS

The first step in developing a spacecraft concept is to define the communications functions that the spacecraft will be required to perform and their range. The spacecraft design concept presented herein provides for the four main types of services originally specified by NOAA and two optional services in which NOAA expressed interest, namely automated field operations and services (AFOS) and imaging.

The operational concepts of the spacecraft are illustrated in figure 1. In this concept, all communications between WSO's are handled through a spacecraft. The use and operation of the spacecraft are controlled by a central coordination station, or control center. The main function of the control center is to control all channel assignments and to demute home warning receivers and spotter transceivers. A WSO issues warn-

[1] The term (WSO) denotes all NOAA weather service offices regardless of hierarchy.

ings and weather reports, coordinates mobile units and other WSO's, and receives data through the spacecraft. The mobile units are capable of two-way voice communication with a WSO. The home receiver is a demutable device capable only of receiving warnings and informative broadcasts. Finally, the data collection platform transmits weather data to WSO's.

Table II specifies the communications links, operating parameters, and communications functions that were used for the sensitivity study. The uplink and downlink frequencies for the warning, mobile reporting, coordinating, and data functions are those recommended in the DWSS feasibility study (ref. 1) performed by the Computer Science Corporation (CSC). To minimize any impact on the DWS, the frequencies for the optional functions of image transmission and AFOS were also selected for 2 gigahertz in order to be consistent with the other DWS communications functions. From table II, it can be seen that the warning function, because of its high power per channel, can have the largest impact on spacecraft power requirements.

To perform a sensitivity study, it was necessary to define a baseline spacecraft. It was decided that the service-intent spacecraft (table I) would be used as a basis for comparing the effects of varying the number of channels of a communications function of the DWS. For bookkeeping reasons, the baseline spacecraft was referred to as spacecraft model II (SM II) during the study and shall be referred to as such in this report.

SPACECRAFT CONFIGURATION

Once the communications functions of the SM II spacecraft were defined, it was possible to design a spacecraft that would provide these communications functions. To minimize initial development costs for a DWS, we decided to use only state-of-the-art technology. As described in the CSC study (ref. 1), attenuation effects in transmitting to low-cost home receivers led to the conclusion that an effective downlink frequency to the home receiver would be 0.79 gigahertz. To efficiently provide local warning coverage at 0.79 gigahertz to the continental United States, Alaska, and Hawaii requires an antenna approximately 9 meters in diameter on the spacecraft. With the requirements of low development cost and a 9-meter antenna, it was evident that a design based as much as possible on ATS-6 (ref. 3) would fit into the DWS concept. This resulted in a spacecraft configuration with an in-orbit weight of 1197 kilograms, a beginning-of-life (BOL) power of 2150 watts, and a cost of $31.4 million.

Table III compares the SM II characteristics with the ATS-6 and Intelsat IV-A characteristics. The SM II has a BOL power three to four times greater than that of ATS-6 or Intelsat IV-A but weighs 204 kilograms less than ATS-6 and 400 kilograms more than Intelsat IV-A. The ATS-6 configuration is heavier than SM II (1) because the ATS-6 employs a rigid nontracking solar array as compared with the SM II flexible array and

(2) because the ATS-6 contains a heavy experimental module (containing the transponders for experiments). The unit cost of SM II in 1974 dollars is $31.4 million, while the Intelsat IV-A cost is $22 million.

The spacecraft configuration that evolved is shown in figure 2 as an artist's concept and in figure 3 as a configuration outline drawing. Readily apparent are the features taken from the ATS-6 (ref. 3), that is, the 9-meter diameter reflector dish, its supporting structure, and the Earth-viewing module (EVM) attached to the end of the reflector support structure. The instrument cubicle contains a fuel tank compartment and two equipment modules: one for the communications system components and the other for the housekeeping components (batteries; power processors; and telemetry and command (T&C), attitude control system (ACS), and reaction control system (RCS) equipment). When the satellite is in orbit, the housekeeping module faces the earth.

A feature of this spacecraft configuration not taken from ATS-6 is the solar array. For these studies, a flexible-substrate, flat-fold array with two wings (ref. 4) was chosen because its design is suitable for the range of BOL power encountered in the study and because this design is convenient for modeling. For Sun tracking, the array is rotated about an axis through the centerline of the wings. Also, to prevent shadowing of the array by the 9-meter reflector dish, each wing is located beyond the reflector perimeter on an extendable mast.

In the configuration outline (fig. 3), the centerline of the RCS fuel tanks is located at the spacecraft's center of gravity in order to eliminate the center-of-gravity shift as the fuel is expended. Also note that the components in the instrument modules are treated as lumped masses and located in a manner to place the spacecraft's center of gravity coincident with the fuel tanks' centerline. These locations determine the overall length of the instrument cubicle. This length plus the length of the reflector structure, which is set by the 4.2-meter reflector focal length, plus the length of structure needed to support the solar array together account for the overall spacecraft height.

SPACECRAFT SENSITIVITY STUDY

Having defined the SM II spacecraft as a baseline for further trade-off studies, it was then possible to proceed with the DWS sensitivity study. The sensitivity study was performed to establish the sensitivity of spacecraft power, weight, and cost by a variation in communications functions (addition or subtraction of channels of fixed bandwidth and radiofrequency power).

To perform the sensitivity study, it was necessary to develop a mathematical model describing the SM II configuration. The mathematical model consisted of equations (1) for determining the required subsystem and total spacecraft power and weight and (2) for estimating the total recurring cost per satellite. It was on this mathematical

model that the computer program used in the sensitivity analysis was based. In deriving the mathematical model, it was assumed that SM II EVM length and solar array size were directly proportional to the communications function (the power into the transponders). A complete explanation of the mathematical model is presented in appendix A. The absolute values of the subsystem parameters were calculated with the assumption that full power is on at all times. This assumption resulted in oversizing the thermal louvered area and would add substantial weight to the spacecraft. The antenna, the truss between the EVM and the antenna, and the upper structure size and weight remained constant and independent of the communications function.

The mathematical model was verified by using it to estimate the weight and BOL power of the SM II configuration. Table IV compares the weight and BOL power as estimated by using the mathematical model and as determined by a detailed analysis on a subsystem level. There is a difference of only 52 kilograms in predicted total spacecraft weight between the two approaches.

The total recurring cost per satellite was based on the cost estimating relation (CER) (ref. 5) explained in appendix A. This cost estimating technique was used by the Goddard Space Flight Center (GSFC) to estimate the cost of the ATS-6 satellite and was shown to be satisfactory.

Once the mathematical model was verified, a computer program was written for calculating the effect on spacecraft power, weight, and cost of variations in communications functions. The digital computer program was run on the Lewis Research Center's IBM 360 time-sharing system. A complete description of the equations and the calculational procedure used in the computer program is presented in appendix A. The variables used in the program are defined in appendix B.

As mentioned previously, the range of communications requirements to be used for the sensitivity study is presented in table I under the column heading "Present spacecraft." However, to analyze all possible combinations of communications functions would require excessive computer runs. Therefore, to limit computer time, only the cases needed to sufficiently establish the sensitivity trends were selected (table V).

DATA REDUCTION AND RESULTS

For bookkeeping reasons, the computer runs were arranged into three categories: those with one warning channel, those with three warning channels, and those with five warning channels. Table VI lists the computer outputs (in order of decreasing BOL power) for various communications functions. For example, table VI(a) shows cost, BOL power, fuel weight, total spacecraft weight, number of batteries required, battery weight, total solar array weight, and radiofrequency power for DWS configurations with a 1-warning-channel and 5-coordinating-channel spacecraft as a reference. From the

computer outputs the average incremental effects of varying communication functions were determined.

The effect of adding a communications function and varying its number of channels consists of the initial cost of that function plus the additional cost of each additional channel for that function. For clarification, the following example is given: The change in cost impact to the DWS for adding data capability was calculated as

$$\text{Total Cost Impact} = \text{IC} + \left(\frac{\text{AC}}{\text{Channel}}\right) \times \text{N} \tag{1}$$

where IC is the initial cost for the first data channel, AC/Channel is the cost for each additional channel after the first, and N is the number of channels.

$$\frac{\text{AC}}{\text{Channel}} = \frac{\Delta C_{200,\,400}}{200} \tag{2}$$

where $\Delta C_{200,\,400}$ is the total cost of going from 200 to 400 data channels

$$\text{IC} = \Delta C_{0,\,200} - \left[199 \times \left(\frac{\text{AC}}{\text{Channel}}\right)\right] \tag{3}$$

where $\Delta C_{0,\,200}$ is the total cost of going from 0 to 200 data channels. Once AC/Channel and IC are determined from equations (2) and (3), respectively, equation (1) may be used to determine the total cost impact for any variation in the number of data channels.

The reason for a cost difference between the first data channel and each additional data channel after the first is that the initial cost of the transponder for data communication is charged to the first data channel. Table VII lists, by communications function, the transponder for each function along with the operating frequency and efficiency of each transponder.

For each communications function, its sensitivity effect on spacecraft cost, weight, and BOL power was determined

(1) By calculating an incremental increase in cost, weight, and BOL power for every case where a function was added

(2) By averaging the incremental increases in cost, weight, and BOL power as determined in step 1 to arrive at an average effect of each additional channel and the initial effect (transponder effect)

Table VIII shows the sensitivity of spacecraft power, weight, and cost to various communications functions. An initial transponder effect is not shown for some of the

functions. The reason that some communication functions were not charged with an initial transponder effect can be seen from table VII. Since all spacecraft configurations analyzed were assumed to have warning and coordination capabilities; the sensitivity effect due to transponders 1 and 2 was charged to the first warning channel and the first coordination channel, respectively. As shown in table VII, the mobile direct and demute functions share the transponder used for the warning function. Since the effect of transponder 1 is charged to the warning function, no initial transponder effect is charged to either the mobile direct or demute functions.

For the values listed in table VIII a function sensitivity study can be performed on a baseline spacecraft such as SM II. Figure 4 shows the cost impact on SM II resulting from either the addition or removal of a selected communications function. From figure 4 it can be seen that the addition of one AFOS channel to the SM II configuration would cost \$0.220 million, as compared with a cost of \$2.2 million for 15 imaging channels.

A similar function sensitivity study was performed on three reference spacecraft derived from the SM II baseline. Table IX lists the three reference cases used to construct sensitivity charts. Figure 5 presents the effect of varying communications functions on the incremental cost, weight, and BOL power of the reference spacecraft defined in table IX. From figure 5, it can be seen that adding five mobile direct and five mobile reporting channels (case A) would cause an increase over a 3-warning-channel, 5-coordinating-channel spacecraft of 130 watts, 40 kilograms, and \$0.875 million. Comparing cases B and E (fig. 5) shows that adding 200 data channels would cause an increase of 139 watts, 34 kilograms, and \$0.77 million. Comparing cases B and A (fig. 5) shows that adding an AFOS channel has a negligible effect on spacecraft cost and weight. However, from cases G, K, and L, adding many imaging channels would have a large effect (33.5 W, 6.4 kg, and \$0.15 million per additional channel) on a DWS.

CONCLUDING REMARKS

This report has described a method for determining the weight, power, and cost sensitivity of the disaster warning satellite (DWS) to variations in broadcasting and fixed services by using a computer model to synthesize the satellite. The technique is applicable to other generic spacecraft when the spacecraft transponder system can be characterized as detailed in appendix A. The analysis of the ATS-6-based DWS resulted in the establishment of "average" impact on satellite weight, power, and cost for all DWS communications functions. The data have been presented in such a way as to facilitate

their use in trade-off studies involving any combination of broadcasting and fixed communications services.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, September 9, 1976,
682-10.

APPENDIX A

MATHEMATICAL MODEL OF DISASTER WARNING SATELLITE

To determine the effect on the spacecraft subsystems of changing the communications requirements is a complex bookkeeping task. A very simplified flow chart is shown in figure 6. For a change in communications power requirements, there will be an attendant change in the power system. The communications and power system changes then force changes in the thermal system, the solar array, and the spacecraft configuration. Changes in weight and solar array area in turn force changes in attitude and stationkeeping fuel requirements. To provide a sound bookkeeping system and a consistent data base from which effects on the spacecraft could be determined, it was decided that a spacecraft synthesizer computer program would be used to study these effects. The program is based on the ATS-6 derived baseline spacecraft. The program and the algorithms are described in this appendix.

Program Description

The program is coded on the Lewis Research Center IBM 360 time-sharing system by using the "namelist" option for program input. The program was written specifically for the configuration based on the ATS-6 and, as such, contains some fixed parameters. The fixed parameters are essentially structural in nature and serve to better identify the spacecraft. The program can be expanded to synthesize other generic spacecraft by altering the fixed parameters. Table X lists the fixed parameters used in the program. Given the fixed parameters in table X and the communications system characteristics, the program synthesizes the second and following changes shown in figure 6.

Communications System

The spacecraft communications system is not directly synthesized in the digital program. Input to the program consists of a dimensioned array of radiofrequency powers, a dimensioned array of transmitter input powers, and the total weight of all transponders. The radiofrequency and input powers are calculated by using an external link analysis program. Table XI shows the channel allocations of the transponders and the power requirements per channel. This basic allocation was used for DWS sensitivity studies. The variations used in the study were the number of channels for a particular function and whether a particular function was offered. As a minimum, trans-

ponders 1, 2, 5, and 8 were assumed for all DWS spacecraft.

Transponder weights in kilograms were calculated as follows:

$$\text{Transponder 1 mass} = 9.09 + (W \times 4.08) + (MD \times 2.27) \tag{A1}$$

where W is the number of active warning channels plus one backup channel and MD is the number of mobile direct channels.

$$\text{Transponder 2 mass} = 3.18 + \frac{MR}{2.2} + (C \times 0.64) \tag{A2}$$

where MR is the number of mobile reporting channels and C is the number of coordinating duplex channels.

$$\text{Transponder 4 mass} = D \times 0.02 \tag{A3}$$

where D is the number of data channels.

$$\text{Transponder 10 mass} = 2.27 + (I \times 0.82) \tag{A4}$$

where I is the number of imaging channels.

$$\begin{aligned}\text{Transponder 3 mass} &= \text{Transponder 5 mass} = \text{Transponder 6 mass} = \text{Transponder 7 mass}\\ &= \text{Transponder 8 mass} = \text{Transponder 9 mass} = 4.09\ \text{kg}\end{aligned} \tag{A5}$$

Equations (A1) to (A5) were used to determine the total communications transponder weight.

Elements of the dimensioned power arrays (for input) are determined from table XI.

Power Subsystem

A conventional power subsystem is assumed, and its configuration is shown in figure 7. The power control unit (PCU) monitors and controls battery charging and discharging operations and controls power flow to the communications and housekeeping power conditioning units (PC's). The power conditioning units provide controlled and regulated power at various voltages to the communications subsystems and the load interface unit. The load interface unit serves as a switching center for the various housekeeping subsystems.

Housekeeping power. - A closed-form solution for the total housekeeping power (including the various loads) can be found from the following design equations:

$$X_1 = (1.0 - \mathrm{PCUE})\mathrm{HSKPW} \tag{A6}$$

where X_1 is the power consumed by the PCU, PCUE is the PCU efficiency, and HSKPW is the total housekeeping power, which includes battery charging power for communicating during an eclipse

$$X_2 = (1 - \mathrm{PCE})\mathrm{HSKPW} \tag{A7}$$

where X_2 is the housekeeping PC power consumption and PCE is the PC efficiency

$$X_3 = \left(\frac{\mathrm{WATEC}}{21.5}\right)\mathrm{BATCH} \tag{A8}$$

where X_3 is the average battery recharging power, WATEC is the eclipse power requirement in watt-hours, BATCH is the battery recharging ratio, and 21.5 is the allowed recharging time in hours

$$X_4 = \mathrm{PLH} \times \mathrm{BHWT} \times \left(\frac{\mathrm{HSKPW} + \dfrac{\mathrm{COMPW}}{\mathrm{PCE}}}{\mathrm{EOLBS}}\right)^{0.5} \tag{A9}$$

where X_4 is the harness power consumption, PLH is the harness power loss factor, BHWT is the base harness weight, COMPW is the total communications transponder input power, and EOLBS is the base end-of-life array power

$$X_5 = \mathrm{LIP} \tag{A10}$$

where LIP is the load interface circuit power (assumed to be constant at 1 W), and

$$X_6 = \text{Constant external loads (ACS, RCS)} \tag{A11}$$

Now the total housekeeping power is given by

$$\mathrm{HSKPW} = \sum_{i=1}^{6} X_i \tag{A12}$$

Note that equation (A9) is a scaling of the harness power consumption based on reference values of harness weight and end-of-life power. In equation (A8) the value of WATEC, the eclipse watt-hour requirement, must be determined.

For design purposes we have assumed that the housekeeping and communications loads are essentially constant during eclipse. However, provision was made to have one of the communications services available for only a portion of the eclipse. In the following equation the service chosen is the warning option:

$$\mathrm{WATEC} = 1.2\left(\mathrm{HSKPW} + \frac{\mathrm{COMPW}}{\mathrm{PCE}}\right) - 1.2(1 - \mathrm{ECLWA})\left(\frac{\mathrm{WARNP}}{\mathrm{PCE}}\right) \tag{A13}$$

where 1.2 is the eclipse time in hours, ECLWA is the percentage of warning capability during an eclipse, and WARNP is the power input to transponders for warning.

We are now in a position to derive the closed-form equation for housekeeping power. Substituting for WATEC in equation (A8) and for X_i in equation (A12) yields an equation of the form

$$A \times \mathrm{HSKPW} + b \times a \times \mathrm{HSKPW} \times \left(\frac{\mathrm{HSKPW} + C}{d}\right)^{0.5} = F \tag{A14}$$

which has the solution

$$\mathrm{HSKPW} = \frac{\left(2AF + \frac{b^2}{d}\right) + \left(4aF\,\frac{b^2}{d} + \frac{b^4}{d^2} + \frac{4a^2cb^2}{d}\right)^{0.5}}{2a^2} \tag{A15}$$

where

$$a = \mathrm{PCE} + \mathrm{PCUE} - 1.0 - 1.2 \times \frac{\mathrm{BATCH}}{21.5}$$

$$b = \mathrm{PLH} \times \mathrm{BHWT}$$

$$c = \frac{\mathrm{COMPW}}{\mathrm{PCE}}$$

$$d = \mathrm{EOLBS}$$

$$F = \frac{BATCH \times 1.2}{21.5 \times PCE} \times [COMPW - (1 - ECLWA) \times WARNP] + X_6 + X_5$$

Batteries. - The program also calculates the number and size of the batteries required for eclipse operation in the following manner: A working number is calculated as

$$B1 = \frac{WATEC}{DOD \times CELAH \times BVMAX} \tag{A16}$$

where WATEC is the eclipse watt-hour requirement, DOD is the allowed depth of discharge, CELAH is the current per battery cell in ampere-hours, and BVMAX is the maximum allowed battery voltage. The number of batteries is found by increasing B1 to the next highest integer. Once the number of batteries is found, the number of cells per battery is determined by iterating from the minimum battery voltage and comparing the total battery capacity, the eclipse watt-hour requirements, and the allowed depth of discharge. A flow chart of this process is shown in figure 8.

Solar array power. - The end-of-life power requirement is given by:

$$EOL = HSKPW + \frac{COMPW}{PCE} \tag{A17}$$

The beginning-of-life power requirement is computed from EOL by allowing for the solstice-pointing offset of the array and the degradation over mission life. The BOL power is computed as

$$BOL = \frac{EOL}{\cos 23.5 \times (1 - DEGRA)} \tag{A18}$$

where DEGRA is the degradation allowance.

Power system weight. - The power system weight is found by summing the following unit weights (in kg):

$$
\left.
\begin{aligned}
&\text{Power control unit weight} = 0.01745 \times \text{Housekeeping power (HSKPW)} \\
&\text{Housekeeping power conditioning weight} = 0.02136 \times \text{HSKPW} \\
&\text{Communications power conditioning weight} = 0.02136 \times \text{Communications power} \\
&\qquad\qquad\text{including efficiencies} \\
&\text{Battery weight} = (\text{Mass/cell}) \times (\text{Number of cells/Battery}) \times \text{Number of batteries} \\
&\text{Harness weight} = \text{BHWT} \times \frac{(\text{EOL})^{0.4}}{\text{EOLBS}} \\
&\text{Load interface weight} = 2.73 \times \frac{(\text{HSKPW})^{0.4}}{489}
\end{aligned}
\right\} \qquad \text{(A19)}
$$

The load interface unit is scaled from the baseline spacecraft design.

Thermal Subsystem

The thermal system design is based on an internal heat pipe/external louver system for two Earth-viewing modules (EVM's) as shown in figure 9. The program design equations compute the louver area, the number of heat pipes, and the amount of insulation required. Basic to the computation is the determination of the internal heat to be dissipated. For the communications module the dissipated power DISP is determined from

$$
\text{DISP} = \frac{\text{COMPW}}{\text{PCE}} - \text{RF} \qquad \text{(A20)}
$$

where COMPW is the total communications transponder input power in watts, PCE is the communications power conditioning efficiency, and RF is the total radiofrequency power output in watts. For the housekeeping module the dissipated power is determined from

$$
\text{HDISP} = \text{HSKPW} - \text{SAOMP} - \text{RCSPW} - \text{CHWAT} + \text{HTCH} \qquad \text{(A21)}
$$

where HSKPW is the total housekeeping power, SAOMP is the solar array orientation mechanism power, RCSPW is the reaction control system power, CHWAT is the battery charging power, and HTCH is the heat loss during battery recharging. Once the

module's heat loss is determined, the thermal system design proceeds identically for both modules.

Louvers. - The required louver area in square meters per face (i.e., north face or south face) is computed from

$$AL = \frac{DISP}{R - ALPHA \times S \times SS - 0.394 \times S \times CS} \tag{A22}$$

where DISP is the communications module dissipated power in watts, R is the design radiation capability in W/m^2, ALPHA is solar absorptance, S is solar input in W/m^2, SS is the sine of the maximum solar incidence angle on a north or south face, and CS is the cosine of the maximum solar incidence angle. Louver height in meters is determined from

$$H = \frac{AL}{1.22} \tag{A23}$$

and the mounting height in meters is given by

$$HM = H + 0.076 \tag{A24}$$

Internal heat pipes. - The number of internal heat pipes is determined by the next largest integer greater than P, where

$$P = \frac{DISP}{40}$$

and we have assumed a heat-pipe load capability of 40 watts. The heat-pipe mounting height in meters is given by

$$PH = \text{Number of heat pipes} \times 0.076 \tag{A25}$$

The effective thermal system height is determined by the maximum of equation (A24) or (A25).

Insulation. - The insulated area is determined by finding the module's external area and subtracting the required louver area.

Weights and costs. - Thermal subsystem component weights and costs are based on the data in table XII. From this data the weight and cost of the module's thermal subsystems are determined.

Calculating the solar array weight involves first determining the size and weight of the array blanket and then calculating the weight of the structure required to support the blanket. Certain basic assumptions are made in these calculations. One is that the array consists of two wings of equal size. This is done to maintain proper dynamic properties of the spacecraft. Another assumption is that the power per unit area of the active blanket is 94.7 W/m^2. The third assumption is that the basic array design is a flat-fold, flexible-substrate type as described in reference 4. With these assumptions, the solar array weight algorithm develops as follows:

(1) Blanket total active area

$$ACTA = \frac{BOL}{PDENS} \tag{A26}$$

where ACTA is the total active area in square meters, BOL is beginning-of-life power in watts, and PDENS is 94.7 W/m^2.

(2) Length of each wing

$$WLEN = \frac{ACTA}{ARWD \times 2} + 2 \times BLNT \tag{A27}$$

where WLEN is the length of the wing in meters, ACTA is the total active area in square meters, ARWD is the array width (1.82 m for these calculations), and BLNT is the blank length at each end (0.15 m for these calculations).

(3) Blanket weight

$$FPW = ACTA \times FLTPC + 4 \times ARWD \times BLNT \times BLWT \tag{A28}$$

where FPW is the flat-pack blanket weight in kilograms, FLTPC is the mass per unit area of the flat-fold blanket (0.858 kg/m^2), BLWT is the substrate density (0.091 kg/m^2).

(4) Extension system weight: If WLEN $\leq$ 10.7 meters, use

$$EXWT = 0.596 \times WLEN \tag{A29}$$

where EXWT is the array extension system mass in kilograms and 0.596 is 2×0.298 kg/m. If WLEN > 10.7 meters, use

$$EXWT = 0.98 \times WLEN$$

where 0.98 is 2×0.49 kg/m.

(5) Guidewire and mechanism weight

$$GWT = WLEN \times 2 \times \left(\frac{1.017}{31} + 2 \times 0.0104\right) \tag{A30}$$

where GWT is the guidewire weight in kilograms, 1.017 kilograms is the mechanism weight for a 31-meter-long array, and 0.0104 kg/m is the guidewire linear density.

(6) Tension system weight

$$TMWT = 2 \times WLEN \times 0.66\left(\frac{0.95}{31} + 2 \times 0.13\right)$$

where TMWT is the tension mechanism weight in kilograms, 0.66 is the ratio of guidewire length to tension wire length, 0.95 kilogram is the mechanism weight for a 31-meter-long array, and 0.13 kg/m is the tension wire weight.

Weight of enclosure cover assembly. - A flat-pack array is launched in a folded configuration, with the enclosure cover applying a force to the folds for preventing movement of the cells, which could result in damage. The weight of the cover and its load-producing mechanism is calculated here. The number of folds in a stowed blanket is

$$FLNO = WLEN \times \frac{1}{0.38} \tag{A32}$$

where 0.38 is the number of meters per fold with WLEN in meters.

The cover preload force in N/m^2 is

$$CUPLD = 15.65 \times FLNO \tag{A33}$$

where 15.65 is a constant involving a safety factor, launch acceleration, panel weight, and the slip resistance between folds; and where FLNO is the number of folds.

The total cover load in newtons is

$$TLD = CUPLD \times 0.38 \times ARWD \tag{A34}$$

where 0.38 meter is the cover width.

The enclosure preload mechanism weight in kilograms is given by

$$ENCLW = TLD \times \frac{9.06}{19\ 082} + 2 \times ARWD \times \frac{6.84}{3.99} \tag{A35}$$

where 9.06/19 080 is the ratio of preload mechanism weight (in kg) to preload force (in N/m^2), and 6.84/3.99 is the ratio of cover weight (in kg) to length (in m) for the reference array.

Container weight for both folded blankets is given by

$$\mathrm{CONW} = 2 \times \mathrm{ARWD} \times \mathrm{FLNO} \times \frac{4.6}{3.9 \times 82} \tag{A36}$$

where 4.6 kilograms is the weight of the reference array container, 3.9 meters is the width of the reference array, and 82 is the number of folds in the reference array.

Total blanket hardware mass in kilograms is given by

$$\mathrm{BLHW} = 2 \times 1.06 \times \frac{\mathrm{ARWD}}{3.9} \tag{A37}$$

where 1.06 kilograms is the blanket hardware weight for the reference array.

The total blanket harness mass in kilograms is given by

$$\mathrm{BLHNW} = 4 \times \mathrm{WLEN} \times \frac{5.85}{31.08} \tag{A38}$$

where 5.85 kilograms is the weight of the harness for the reference array, 4 denotes two wings with harness on two edges, and 31.08 meters is the length of the reference array.

The total solar array weight in kilograms is

$$\mathrm{SAW} = \mathrm{FPW} + \mathrm{GWT} + \mathrm{TMWT} + \mathrm{ENCLW} + \mathrm{CONW} + \mathrm{BLHW} + \mathrm{BLHNW} + \mathrm{EXWT} + 1.6 \tag{A39}$$

where 1.6 kilograms is the weight of the following fixed items:

(1) Mast tip fitting, 0.68 kilogram

(2) Tension hardware, 0.95 kilogram.

EVM Layout Design

The major impetus in the design of the Earth-viewing module (EVM) is the requirement that the center of gravity (c.g.) of the spacecraft be located on the centerline of the fuel tanks. From figure 3 and table X, we note that most of the components "above" the c.g. are fixed in mass (i.e., reflector, truss structure, upper structure, solar ar-

ray deployment, etc.). The lone exception is the solar array. In contrast, the components "below" the c.g. are variable in mass (i.e., power subsystem, thermal subsystem, transponders, and structure). To maintain the c.g. on the centerline of the tanks, we shifted the variable-mass components in the EVM to keep the fixed-mass moments "above" and "below" the c.g. equal. The method chosen to accomplish this is described here.

A sketch of the EVM layout is given in figure 10. From top to bottom there is a fixed section containing the fuel tanks, a communications module containing the transponders, and a housekeeping module containing the ACS, T&C, power, and other subsystems. The ACS, T&C, power (excluding the harness), and communications subsystems are considered as lumped mass parameters spaced at fixed intervals in the EVM. The fixed intervals are determined by the baseline spacecraft design. In addition, the housekeeping module structure and its thermal subsystem are considered as lumped mass parameters located at the center of the housekeeping module, and the communications module structure and its thermal subsystem are considered as lumped mass parameters located at the center of the communications module. The EVM electrical harness is considered to be a distributed mass parameter.

The design variable is X, the distance between the top of the communications module and the first lumped transponder mass. As X varies, the communications module length changes and the housekeeping module shifts with respect to the c.g. By equating the mass moments "above" and "below" the c.g. and solving for X, we have an estimate of the EVM layout.

The total mass moment above the c.g., in kg · m, is given by

$$M_1 = SAW(5.7) + 1228.23 \tag{A40}$$

where SAW is the solar array mass; 5.7 is the fixed moment arm in meters from the c.g. to the solar array centerline; and $1228.23 = \sum_i m_i l_i$, where the m_i and l_i are given in table X. The mass moments below the c.g., in kg · m, are given by the following design equations (refer to fig. 10 for moment arm determination):

(1) Communications module mass moments

$$M_2 = TWC\left(\frac{X}{2} + 0.82\right) \tag{A41}$$

where TWC is the communications module thermal system mass in kilograms,

$$M_3 = \frac{COMWT}{2} (X + 0.37) \tag{A42}$$

where COMWT is the communications transponder mass in kilograms,

$$M_4 = \frac{COMWT}{2} \times (X + 0.98) \tag{A43}$$

$$M_5 = 23.1 \times \left[X + 0.91 \times \left(\frac{X}{2} + 0.82\right)\right] \tag{A44}$$

where 23. 1 kilograms is the mass per linear meter of the structure.

(2) Housekeeping module mass moments

$$M_6 = HTWC \times (X + 2.19) \tag{A45}$$

where HTWC is the housekeeping module thermal subsystem mass in kilograms,

$$M_7 = 43.09 \times (X + 2.19) \tag{A46}$$

where 43. 09 kilograms is the housekeeping module structural mass,

$$M_8 = (ACS + TCWT) \times (X + 2.5) \tag{A47}$$

where ACS is the attitude control system mass in kilograms excluding fuel tanks and TCWT is the telemetry and command system mass in kilograms,

$$M_9 = PS \times (X + 2.8) \tag{A48}$$

where PS is the power system mass in kilograms excluding the harness, and

$$M_{10} = 11.36 \times (X + 3.11) \tag{A49}$$

where 11. 36 kilograms is the mass of the small antennas and mountings.

(3) Miscellaneous mass moments

$$M_{11} = 11.36 \times (X + 1.28) \tag{A50}$$

where 11. 36 kilograms is the mass of the module attachment hardware, and

$$M_{12} = HWM \times \left(\frac{X}{2} + 1.74\right) \tag{A51}$$

where HWM is the mass of the electrical harness in kilograms, distributed in the EVM.

The total mass moment below the c. g. is given by

$$M_{13} = \sum_{i=2}^{12} M_i \tag{A52}$$

Note that, from equation (A44), M_{13} will be quadratic in X. Expanding M_{13} and equating to M_1 give

$$11.56\,X^2 + X\left(95.41 + \frac{TWC}{2} + COMWT + HTWC + ACS + TCWT + PS + \frac{HWM}{2}\right) + 161.49$$

$$+ 0.82\,TWC + 0.68\,COMWT + 2.19\,HTWC + 2.5(ACS + TCWT) + 2.8\,PS + 1.74\,HWM$$

$$= 5.7\,SAW + 1228.23 \tag{A53}$$

Equation (A53) may be solved for X, the design length. From knowledge of X we compute the EVM length (EVML), the final structure weight, and the datum locations of the lumped subsystems from the c. g.

Fuel Requirements

The attitude control and reaction control subsystems' dry weight and power requirements are inputs to the program. The ACS weight and power are derived from the ATS-6 system, which is a three-reaction-wheel system. The RCS's weight and power are also derived from ATS-6 with additional redundant hydrazine thrusters for north-south stationkeeping. No attempt is made in the program to optimize fuel tankage. Rather it was decided that two existing, flight-proven tanks capable of holding in excess of 230 kilograms of hydrazine would be used in the design. These tanks could be off loaded as necessary.

Attitude control fuel. - The attitude disturbance torques are computed only for solar pressure effects. The solar arrays are assumed to be the only structural elements that are affected. Figure 11 defines a coordinate system and the location of the attitude control thrusters. Note that a large 5.7-meter moment arm exists between the spacecraft center of mass and center of pressure. Since the solar arrays track the right ascension of the Sun, a sinusoidal torque about the pitch axis is produced. The period of the sinusoid is 1 day. The required momentum storage about the pitch axis in N · m/sec is

given by

$$HY = 5.7 \times AR \times 0.17122 \tag{A54}$$

where AR is the total array area in square meters, and

$$0.17122 = \int_0^{1/2\,\mathrm{day}} F(1 + RFL)SLN\, W_0 t\, dt$$

where F is the solar force (4.788×10^{-6} N/m^2), RFL is the solar array reflectivity (0.3), and W_0 is the orbit rate (7.27×10^{-5} rad/sec).

Torque in the roll-yaw plane is also produced by solar pressure. This torque is due to the shear force component of the solar pressure force. The maximum roll-yaw torque in N/m is given by

$$TXMAX = AR \times CS \times SS \times 3.352\times10^{-6} \times 5.7 \tag{A55}$$

where CS and SS are the cosine and sine, respectively, of the maximum solar incidence angle, and

$$3.352\times10^{-6} = (1 - R) \times 4.788\times10^{-6}$$

Fuel requirements are computed as follows:

$$BUPF = \frac{HY \times 2 \times 5 \times 365}{AM \times SPP} \tag{A56}$$

where we have allowed the pitch wheel to fail immediately and where BUPF is the backup pitch fuel, AM is the moment arm to the thrusters and equals (EVML - 0.36 meter), and SPP is the thruster specific impulse for pulsed operation in N · sec/kg.

$$ACFX = \frac{HX \times 2 \times 5 \times 365}{3 \times SPP \times AM} \tag{A57}$$

where ACFX is the roll-yaw momentum dumping fuel and

$$HX = TXMAX\ N/m \times 86\,400\ sec/day$$

The 2/3 factor in equation (A57) allows for a less-than-maximum average solar incidence

angle. In addition, a nominal fuel allowance is computed for pitch-axis momentum dumping:

$$ACFY = \frac{AR \times 6.224\times10^{-6} \times 86\ 400 \times 5 \times 365}{SPP \times AM} \tag{A58}$$

This allowance is based on an equivalent 0.3-meter constant separation of the center of pressure and center of mass. The total attitude control fuel is given by

$$TFF = ACFX + ACFY + BUPF \tag{A59}$$

Stationkeeping fuel. - The stationkeeping fuel is based on the spacecraft dry weight and the average fuel load over the mission life. The total stationkeeping fuel is given by

$$STKF = \frac{\left(SCDWT + \frac{TFF}{21}\right) \times DELV}{SPC \times \left(1 - \frac{DELV}{2 \times SPC}\right)} \tag{A60}$$

where SCDWT is the spacecraft dry weight in kilograms, SPC is the thruster specific impulse for continuous operation in N · sec/kg, and DELV is the delta velocity in m/sec required for north-south and east-west stationkeeping for the mission life.

Spacecraft weight. - The spacecraft launch weight can now be computed from

$$SCW = SCDWT + TFF + STKF \tag{A61}$$

Spacecraft Cost

To determine the impact of a spacecraft design change, it was necessary to find a reasonable method of estimating the unit cost (recurring cost) of the spacecraft. The method chosen was to use a cost estimating relation (CER) based on the spacecraft weight and power. The cost in millions of dollars is given by

$$COST = \frac{\left\{\left[0.2454(SCDWT)^{0.42971}\right] \times \left[(Power)^{0.17339}\right] \times \left[(EU)^{0.95632}\right]\right\}}{EU} \tag{A62}$$

where SCDWT is the spacecraft dry mass in kilograms, power denotes the end-of-life (EOL) power in watts, and EU is the equivalent unit of spacecraft (4.6 for DWS). This

equation was developed in reference 6 and has been updated to 1974 dollars. In reference 6, spacecraft programs resulting in smaller, less-powerful spacecraft were used to determine the CER. Therefore, some justification for the use of the CER must be provided.

A comparison was made between the CER and the engineering cost method shown in table XIII. The comparison was based on the service-intent spacecraft. First, the basic spacecraft cost was computed by the two methods. Second, the cost of one additional warning channel was computed. The comparison is summarized in table XIV. Since the increased costs for the additional warning channel compared favorably, it was decided to use the CER method (eq. (A62)).

In equation (A62), the term EV requires further explanation. The development of the equivalent units procedes as outlined in reference 6 and is summarized in table XV. For this range of EU's,

$$\frac{(EU)^{0.95632}}{EU} = 0.94 \tag{A63}$$

It can be seen from equation (A63), that the EU's have a small impact in determining the spacecraft unit (or recurring) costs.

APPENDIX B

DISASTER WARNING SATELLITE SPACECRAFT SYNTHESIZER

COMPUTER PROGRAM VARIABLES

Input

Variable	Definition
ACSPW	attitude control system power, W
ACSWT	attitude control system mass (dry), kg
ALNMS	maximum solar array - Sun misalinement, deg
ALPHA	absorptance
ARWD	array width, m
BATCH	battery recharging ratio, W_{in}/W_{stored}
BHWT	base harness mass, kg
BLNT	blank array panel length at each end of array, m
BLWT	substrate density, kg/m^2
BVMIN	minimum battery voltage, V
BVMAX	maximum battery voltage, V
CELAH	current density per cell, A-hr
CELLV	battery cell voltage, V
COMP(I)	power for array of communications system items, W
COMPRF(II)	radiofrequency output power for array of communications
COMWT	communications transponder system mass, kg
DEGRA	solar array degradation allowance (0 < DEGRA < 1)
DOD	maximum depth of discharge (0 < DOD < 1)
ECLWA	factor of time for warning during eclipse (0 ≤ ECLWA ≤ 1.0)
EOLBS	base end-of-life array power, W
EU	equivalent units of spacecraft
FLTPC	density of flat-fold array active area, kg/m^2
PCE	power conditioning efficiency

Variable	Definition
PCUE	PCU efficiency
PDENS	solar array power density, W/m^2
PLH	harness power loss factor
R	heat lost to space, W/m^2
RCSPW	reaction control system power, W
RCSWT	reaction control system mass, kg
RFL	reflectivity
RLUP	rollup array active area density, kg/m^2
S	solar heating constant, W/m^2
SAOMP	solar array orientation mechanism power, W
SPC	thruster specific impulse for continuous firing, $N \cdot sec/kg$
SPP	thruster specific impulse for pulsed firing, $N \cdot sec/kg$
TCPW	telemetry and command system power, W
TCWT	telemetry and command system mass, kg

Output

AC	communications module louver cost, thousands of dollars
ACF	attitude control fuel weight (not stationkeeping), kg
ACFX	attitude control fuel weight (roll-yaw plane), kg
ACFY	attitude control fuel weight (pitch), kg
ACS	lumped mass used in c. g. calculations, kg
ACTA	total solar array active area, m^2
ADOD	actual depth of battery discharge
ALC	communications module louver area, m^2
AM	thruster moment arm, m
AMLI	communications module multilayer-insulation (MLI) area, m^2
AR	total solar array area, m^2
AW	communications module louver mass, kg

Variable	Definition
BATNO	number of batteries
BATW	battery mass, kg
BLHNT	blanket harness mass, kg
BLHW	blanket hardware mass, kg
BOL	beginning-of-life array power, W
BUPF	backup pitch fuel, kg
BV	battery voltage, V
CAP	battery capacity (total), W-hr
CELNO	number of cells per battery
CHWAT	average battery charging power, W
CMLI	cost of communications module multilayer insulation, thousands of dollars
CMODL	communications module length based on c.g. calculations, m
CMODW	communications module mass (structure only), kg
COMPW	total communications system power without conditioner efficiency, W
CONW	solar array container mass, kg
COST	spacecraft unit cost, millions of dollars
CUPLD	solar array cover preload, N/m^2
DELV	stationkeeping delta velocity, m/sec
DISP	communications module dissipated power, W
D1	distance of communications module thermal system from c.g., m
D2	distance of first section of communications transponder from c.g., m
D3	distance of housekeeping module thermal system from c.g., m
D4	distance of lumped ACS weight from c.g., m
D5	distance of lumped power system weight from c.g., m
D6	distance of distributed harness weight from c.g., m
ENCLW	array enclosure mass, kg
EOL	end-of-life array power, W
EVML	total EVM length based on c.g. calculations, m
EXWT	array extension system mass, kg

Variable	Definition
FLNO	number of folds per wing for flat fold
FPW	total flat-fold blanket mass, kg
GWT	guidewire mass, kg
H	communications module louver height, m
HAC	housekeeping module louver cost, thousands of dollars
HALC	housekeeping module louver area per side, m^2
HARNW	harness mass, kg
HAMLI	housekeeping module MLI area, based on thermal calculations, m^2
HAW	housekeeping module louver mass, kg
HCMLI	housekeeping module MLI cost, thousands of dollars
HDISP	housekeeping module dissipation power, W
HH	housekeeping module louver height, m
HHM	housekeeping module louver mounting height, m
HM	communications module louver mounting height, m
HOSRC	housekeeping module mirror cost, thousands of dollars
HOSRW	housekeeping module mirror mass, kg
HP	number of housekeeping module pipes
HPC	housekeeping module pipe cost, thousands of dollars
HPH	housekeeping module pipe mounting height, m
HPW	housekeeping module pipe mass, kg
HSKPW	total housekeeping power, W
HSKWT	housekeeping power-conditioner mass, kg
HTCC	housekeeping module thermal system cost, thousands of dollars
HTCH	heat loss during battery recharging, W
HTMH	housekeeping module height based on thermal calculations, m
HTWC	housekeeping module thermal system mass, kg
HWM	distributed harness mass, kg
HWMLI	housekeeping module MLI mass, kg
HX	maximum roll-yaw momentum storage, N · m/sec

Variable	Definition
HY	maximum pitch momentum storage, N · m/sec
OSRC	communications module mirror cost, thousands of dollars
OSRW	communications module mirror mass, kg
P	number of communications module pipes
PC	communications module pipe cost, thousands of dollars
PCCOM	communications module power-conditioner mass, kg
PCUWT	PCU mass, kg
PH	communications module pipe mounting height, m
PL	subsystem power load (ACS, RCS, T&C, SAOM), W
PS	lumped power system mass, for c.g. calculations, kg
PSWT	total power system mass, kg
PW	communications module pipe mass, kg
RF	total radiofrequency power output, W
RLUPW	rollup array blanket mass, kg
SAW	total solar array mass, kg
SCDWT	spacecraft dry mass, kg
SCW	total spacecraft mass, kg
SIG1	mass moment above c.g., kg · m
STKF	total stationkeeping fuel, kg
TCC	communications module thermal system cost, thousands of dollars
TF	total fuel, kg
TFF	total nonstationkeeping fuel, kg
TLD	total solar array cover load, N
TMH	communications module thermal system height, m
TMWT	array tension mechanism mass, kg
TTC	total thermal system cost, thousands of dollars
TTW	total thermal system mass, kg
TWC	communications module thermal system mass, kg
TXMAX	maximum roll-yaw torque, N/m

Variable	Definition
ULI	load interface unit mass, kg
WARNP	power for warning without power-conditioning efficiency (= COMP(1)), W
WATEC	capacity required for eclipse operation, W-hr
WLEN	solar array wing length, m
WMLI	communications module MLI mass, kg
X	communications module variation (X + 0.91 = Length of module), m

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TABLE I. - EVOLUTION OF SPACECRAFT

COMMUNICATIONS REQUIREMENTS

Function	Initial spacecraft (1972)	Service-intent spacecraft (1974) (SM II)[a]	Present spacecraft
	Number of channels		
Warning	10	3	1 - 5
Mobile direct	5	10	5 - 10
Mobile reporting	50	25	5 - 25
Data collecting	200	200	0 - 400
Duplex voice - coordinating	5	5	5 - 10
AFOS[b]	0	0	0 - 1
Imaging	0	0	0 - 50

[a]Spacecraft model II (baseline spacecraft).

[b]Automated field operations and services.

TABLE II. - COMMUNICATIONS FUNCTIONS OF DISASTER WARNING SATELLITE

From -	Frequency, GHz		Type of function	To -	Spacecraft radiofrequency power per channel, W	Function	Type
	Uplink	Downlink					
Weather service office	6	0.79	Required	Public	60	Warning	Voice
Weather service office	6	.79		Spotters	1.2	Direction	Voice
Control center	6	.79		Public	4	Demute	Signal
Spotter	2	2		Weather service office (WSO)	1.6	Reporting	Voice
Spotter				Control center	.5	Channel requisition	Digital
Weather service office				Control center	2	System control	Digital
Control center				Spotter	1	Channel assignment	Digital
Control center				WSO	1	System control	Digital
Control center				WSO, spotter	4	Pilot tone	-------
Weather service office, control center				WSO, control center	1.6	Coordination	Voice
Data collection platform	.4			WSO	.03	Data	Digital
Control center	2	.468		Data collection platform	2	Interrogate	Digital
Weather service office	2	2	Optional	WSO	6	Imaging	-------
Weather service office	2	2	Optional	WSO	6	AFOS	-------

TABLE III. - COMPARISON OF SPACECRAFT CHARACTERISTICS

Characteristics	Spacecraft model II (SM II)[a]	ATS-6	Intelsat IV-A
Beginning-of-life power, W	2186	650	500
Weight, kg	1197.5	1401.6	798.3
Cost, millions of dollars	31.4	------	~22

[a]Baseline spacecraft (service-intent spacecraft).

TABLE IV. - COMPARISON BETWEEN MATHEMATICAL MODEL AND ANALYSIS BY SUBSYSTEM LEVEL FOR SPACECRAFT MODEL II CONFIGURATION

Characteristics	Mathematical model	Analysis
Beginning-of-life power, W	2186	2150
Weight, kg	1197.5	1249.65

TABLE V. - VARIATIONS IN COMMUNICATIONS FUNCTIONS SELECTED FOR SENSITIVITY STUDIES - SPACECRAFT MODEL II CONFIGURATION

Service	Number of channels
Warning	1, 3, 5
Mobile direct	0, 5, 10
Mobile reporting	0, 5, 15, 25
Data collecting	0, 200, 400
Duplex coordinating	5, 10
AFOS[a]	0, 1
Imaging	0, 15, 30, 50

[a]Automated field operations and services.

TABLE VI. - OUTPUT OF COMPUTER RUNS

(a) Reference case: one warning channel, five coordinating channels

Computer run	Coordination (Number of channels)	Mobile direct (Number of channels)	Mobile reporting (Number of channels)	AFOS[a] (Number of channels)	Data (Number of channels)	Imaging (Number of channels)	Cost, millions of dollars	Beginning-of-life power, W	Fuel weight, kg	Total spacecraft weight, kg	Number of batteries	Battery weight, kg	Solar array weight	Radio-frequency power, W
31						50	33.47	2770.00	185.47	1270.7	5	114.76	45.89	388.5
28						30	30.622	2101.5	161.30	1147.68	4	87.82	37.09	268.5
34		5	5	1	400	15	30.71	2039.1	163.34	1168.68	3	83.82	36.27	230.5
10		10	25				28.58	1601.0	147.60	1085.22		65.86	30.51	140.5
25						15	28.17	1599.3	142.97	1049.57		65.86	30.49	178.5
22		5	5	1	400		28.24	1536.8	145.15	1071.7		62.87	29.66	124.5
19		5	5	1	200		27.50	1397.7	140.39	1045.53		59.87	27.84	116.5
7		5	15				27.42	1389.9	139.7	1040.63		59.87	27.73	118.5
16	5	5	5	1			27.5	1414.8	139.98	1041.18		59.87	28.06	124.5
13	-	5	5	1			26.605	1259.0	134.17	1008.65	2	51.89	26	108.5
4	-	5	5				26.37	1225.0	132.5	998.63	2	49.9	25.56	102.5
1			Reference case				25.32	1096.0	124.87	948.96	2	45.9	23.86	88.5

(b) Reference case: three warning channels, five coordinating channels

Computer run	Coordination	Mobile direct	Mobile reporting	AFOS[a]	Data	Imaging	Cost, millions of dollars	Beginning-of-life power, W	Fuel weight, kg	Total spacecraft weight, kg	Number of batteries	Battery weight, kg	Solar array weight	Radio-frequency power, W
32						50	35.172	3215.2	201.44	1348.3	5	134.72	51.74	508.5
29						30	32.462	2547.3	176.22	1221.61	4	103.78	42.96	388.5
35		5	5	1	400	15	32.628	2484.9	178.94	1247.52	4	103.78	42.14	350.5
SM II	5	10	25		200		31.410	2186.2	168.96	1198.39	4	91.807	38.20	268.5
11		10	25				30.668	2047.5	162.61	1162.87	3	83.82	36.38	260.5
26						15	30.251	2045.7	157.94	1127.18		83.82	36.36	298.5
23		5	5	1	400		30.324	1983.3	159.80	1146.73		80.83	35.54	244.5
20		5	5	1	200		29.705	1844.5	155.36	1124.0		77.56	33.71	236.0
8		5	15				29.50	1836.7	153.13	1108.04		74.84	33.61	238.0
17	5	5	5	1			29.70	1861.6	188.88	1119.56		77.84	33.94	244.0
14		5	5	1			28.938	1705.7	149.37	1090.03		71.85	31.887	228.5
5		5	5				28.684	1672.2	147.19	1076.19		69.86	31.446	222.5
2			Reference case				27.78	1543.4	139.84	1030.29		62.87	29.75	208.0

(c) Reference case: five warning channels, five coordinating channels

Computer run	Coordination	Mobile direct	Mobile reporting	AFOS[a]	Data	Imaging	Cost, millions of dollars	Beginning-of-life power, W	Fuel weight, kg	Total spacecraft weight, kg	Number of batteries	Battery weight, kg	Solar array weight	Radio-frequency power, W
33						50	36.77	3660.1	216.64	1423.64	6	149.69	61.80	628.5
30						30	33.667	2992.6	185.75	1252.1	5	124.74	48.816	508.5
36		5	5	1	400	15	34.348	2930.4	194.0	1320.72	5	119.75	47.996	470.5
12		10	25				32.58	2493.3	178.26	1241.66	4	103.78	42.248	380.5
27						15	32.164	2491.6	173.54	1205.97		103.78	42.22	418.5
24		5	5	1	400		32.28	2429.2	175.59	1227.38		99.79	41.405	364.5
21		5	5	1	200		31.64	2290.6	170.1	1198.03		95.75	39.58	356.5
9		5	15				31.512	2282.8	168.74	1188.46		95.80	39.478	358.5
18	5	5	5	1			31.626	2307.6	169.69	1193.54		95.80	39.805	364.5
15		5	5	1			30.909	2152.0	163.75	1162.15		87.815	37.76	348.5
5		5	5				30.745	2118.5	162.39	1154.76		87.815	37.317	342.5
3			Reference case				29.90	1990.0	154.95	1108.76	3	83.82	35.626	328.5

[a]Automated field operations and services.

TABLE VII. - SPACECRAFT TRANSPONDER ALLOCATION PLAN

Function	Transponder									
	1	2	3	4	5	6	7	8	9	10
	Transponder efficiency, percent									
	55	35	20	20	20	20	20	55	35	35
Warning	●									
Mobile direct	●									
Demute	●									
Coordination (duplex)		●								
Mobile reporting		●								
Data collection platform interrogate			●							
Data collection platform				●						
System control					●					
Channel request						●				
Channel assignment							●			
Pilot tone								●		
Automated field operations and services									●	
Imaging										●
Uplink frequency, GHz	6	2	2	2	2	2	2	2	2	2
Downlink frequency, GHz	0.79	2	0.468	2	2	2	2	2	2	2

TABLE VIII. - SENSITIVITY OF SPACECRAFT TO VARIOUS COMMUNICATIONS FUNCTIONS[a]

Function	Averaged initial transponder effect on spacecraft -			Averaged effect of each additional channel on spacecraft -		
	Power, W	Weight, kg	Cost, thousands of dollars	Power, W	Weight, W	Cost, thousands of dollars
Warning	----	-----	---	223.2	38.69	1100
Automated field operations and services	33.5	10.43	220	-----	-----	----
Imaging	33.5	10.43	220	33.5	6.44	155
Coordinating	----	-----	---	31.2	6.21	160
Mobile direct	----	-----	---	9.4	4.08	80
Mobile reporting	----	-----	---	16.5	3.9	95
Data	.7	10.07	123	.7	.127	3

[a]Reference cases have warning and coordinating capability.

TABLE IX. - REFERENCES CASES

[Five coordination channels.]

	Number of warning channels		
	1	2	3
Total spacecraft weight, kg	948.96	1030.29	1108.76
Total spacecraft cost, millions of dollars	25.32	27.79	29.90
Total spacecraft beginning-of-life power, W	1096	1543	1990

TABLE X. - FIXED SPACECRAFT PARAMETERS

Spacecraft design life, yr 5
Earth-viewing module cross section, m . . . . 1.37 × 1.37
Solar array reflectivity 0.3

Fixed structure

Item	Mass, kg	Distance from center of gravity, m
9.14-Meter reflector	86.36	4.48
Upper structure	36.36	5.70
Reflector support truss	109.09	2.50
Solar array deployment system	50.0	5.70
Solar array orientation and power transfer system	11.82	5.70
Truss thermal system	2.27	2.50
9.14-Meter-reflector feed system	9.09	.37

TABLE XI. - COMMUNICATIONS SYSTEM

Transponder	Function	Radio-frequency power per channel, W	Input power per channel, W	Bandwidth
1	Warning	60	114	16 kHz
1	Mobile direct	1.2	4.74	15 kHz
1	Demute	4	5.25	100 bps
2	Mobile reporting	1.6	8.4	16 kHz
2	Coordinating (voice, duplex)	3.2	15.92	16 kHz
3	Data collection platform interrogate	2.0	------	40 bps
4	Data collection platform data	.03	------	100 bps
5	System control 1	1.0	13.1	4400 bps
	System control 2	2.0		7260 bps
6	Channel requisition (mobile)	.5	------	200 bps
7	Channel assignment (mobile)	1.0	------	100 bps
8	Pilot tone	4.0	------	Continuous wave
9	Automated field operations and services	6.0	17.15	25 kbs
10	Imaging	6.0	17.15	25 kbs

TABLE XII. - THERMAL SUBSYSTEM

Component	Weight factor	Cost factor
Louver	5.27 kg/m^2	\$48.44×$10^3/m^2$
Heat pipes	1.77 kg/pipe	\$5×10^3/pipe
Insulation	1.71 kg/m^2	\$19.48×$10^3/m^2$
Reflectors mounted beneath louvers	1.137 kg/m^2	\$5.38×$10^3/m^2$

TABLE XIII. - ENGINEERING COST METHOD ASSUMING PRIME CONTRACTOR APPROACH

Assembly cost	0.015 × Subsystem hardware cost (SHC)
Spacecraft integration, test, and evaluation cost (IT & E)	0.21 × SHC
Direct cost (DC)	SHC + Assembly cost + IT & E
General and administrative cost (G & A)	0.136 × DC
Fee	0.07 × DC
Research and development cost (R & D)	DC + G & A + Fee
Contingency	0.15 × R & D
Management cost	0.1 × R & D
Total cost	R & D + Contingency + Management

TABLE XIV. - COMPARISON OF COST METHODS FOR SERVICE-INTENT SPACECRAFT

	Engineering cost method	Cost estimating relation (CER)
	Cost, millions of dollars	
Subsystem hardware	15	(a)
Total	27.75	31.41
One additional warning channel	.98	1.02

[a]Based on spacecraft weight of 1022 kg and power of 1427 W.

TABLE XV. - DISASTER WARNING SYSTEM EQUIVALENT UNITS FOR ATS-6 INHERITANCE

	Equivalent units (EU)
Design and development	0.9 - 1.2
Thermomechanical test unit	.2
Engineering test unit	.7
Protoflight (flight A)	1.8
Nonrecurring total	3.6 - 3.9
Flight B	1.0
Total	4.6 - 4.9

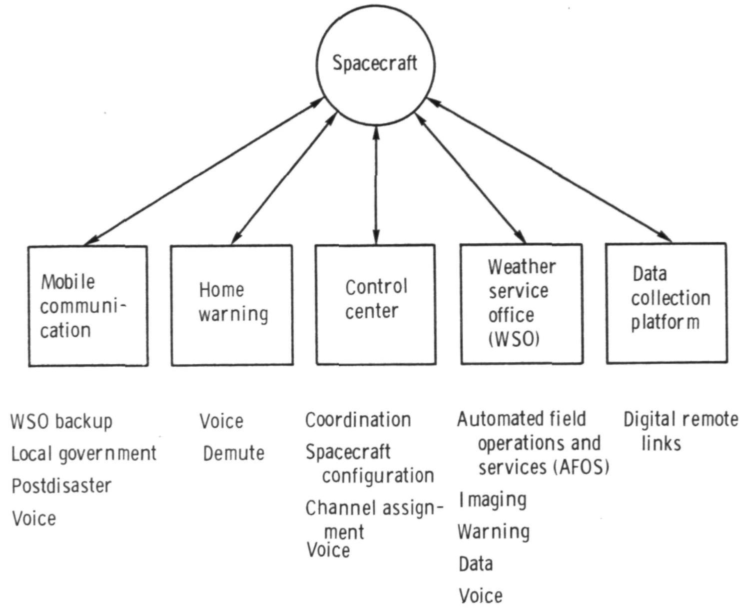


Figure 1. - Operational concepts for disaster warning satellite.

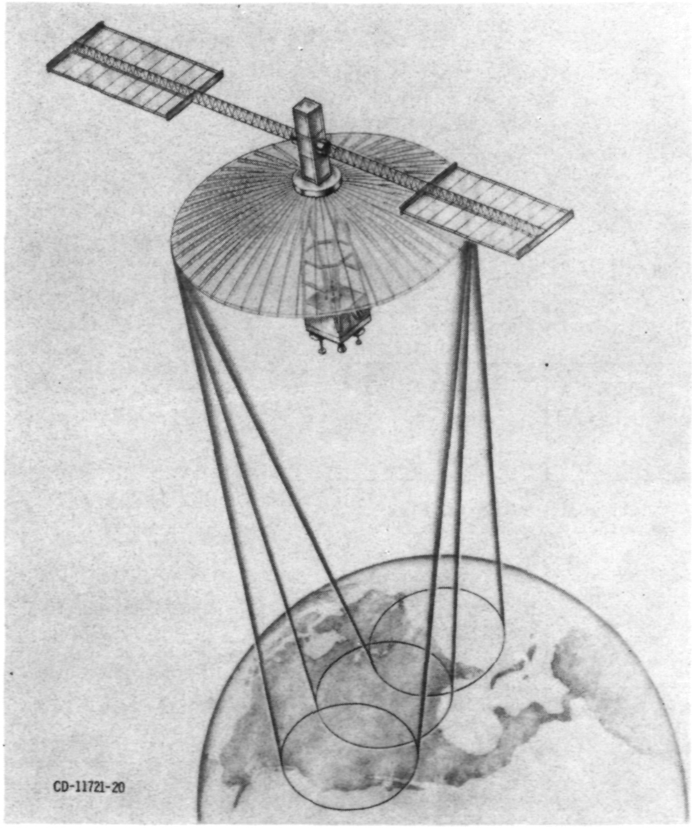


Figure 2. - Disaster warning satellite - spacecraft model II configuration.

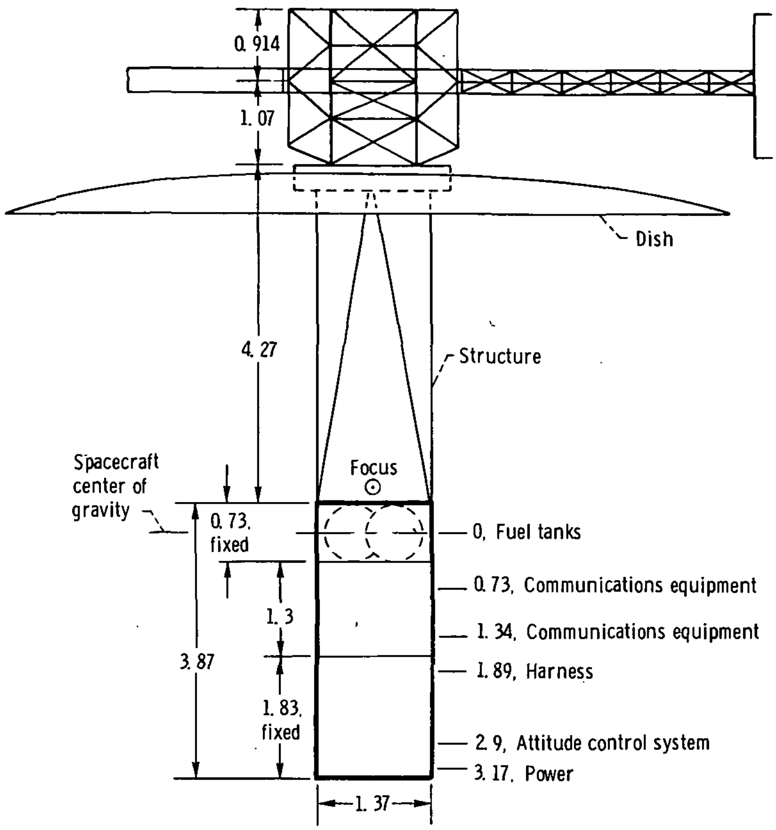


Figure 3. - Configuration outline of disaster warning spacecraft - spacecraft model II configuration. (Dimensions are in meters.)

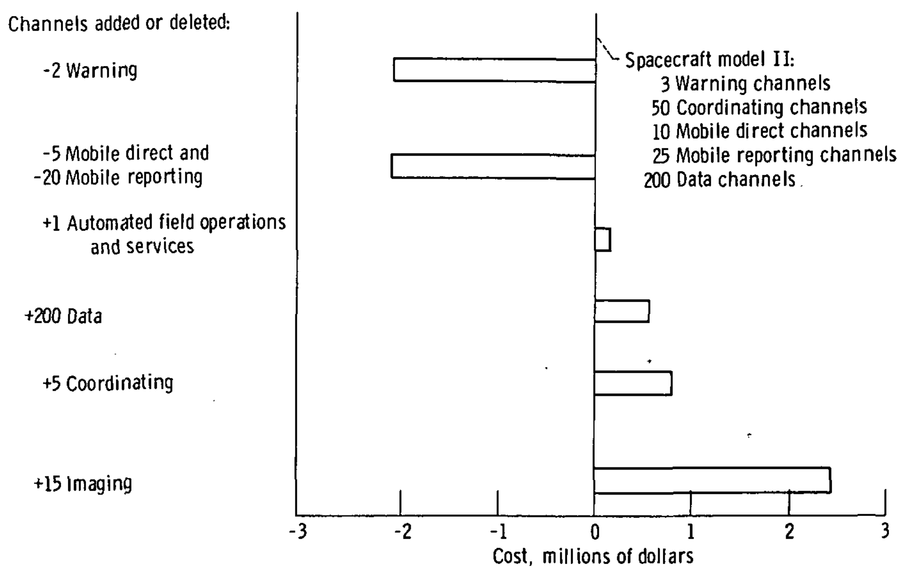
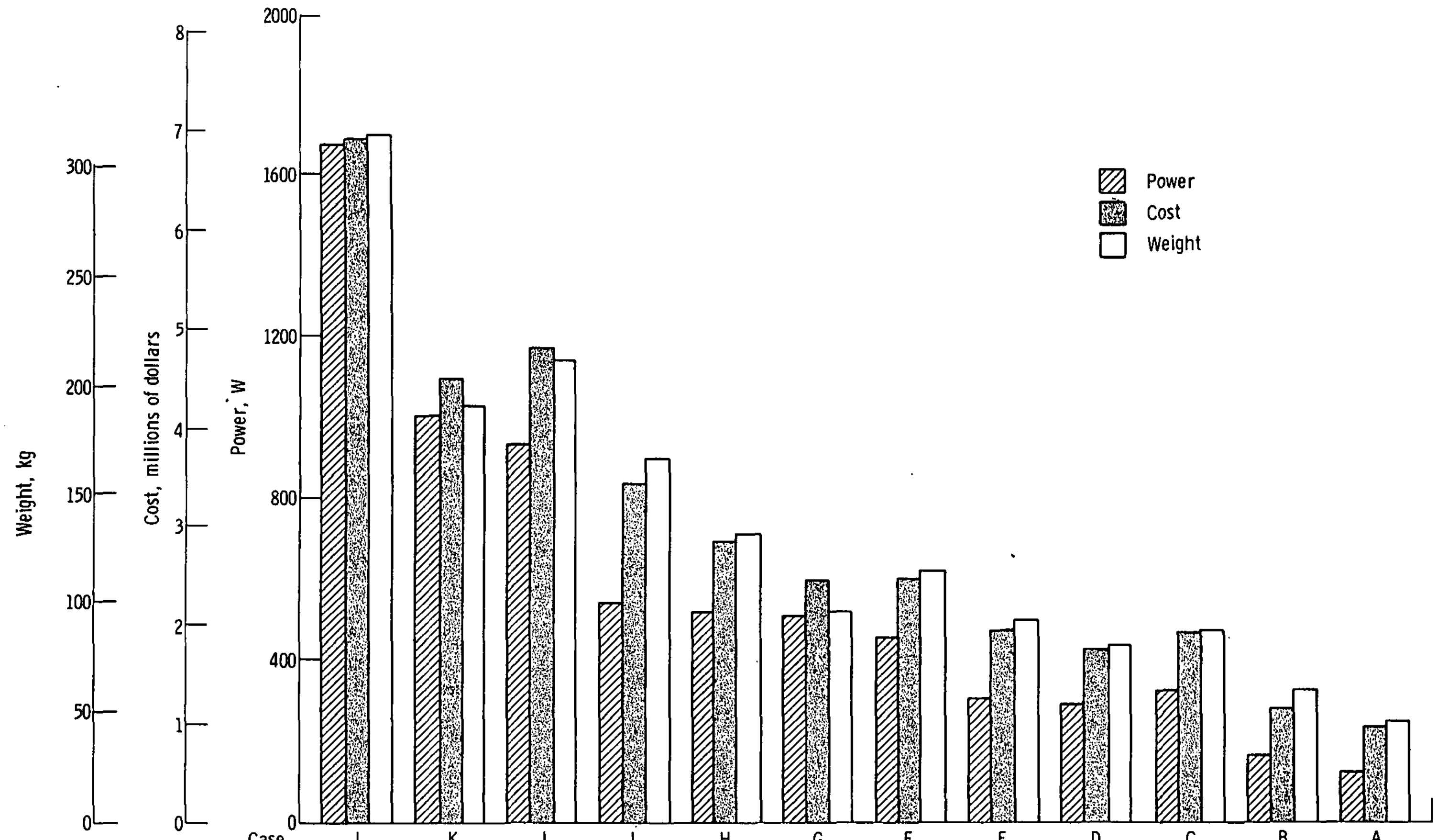


Figure 4. - Effect of adding or deleting communications functions on cost of spacecraft model II.



Case	L	K	J	I	H	G	F	E	D	C	B	A
Additional channels: Imaging	50	30	15			15						
Data			400	200			400	200				
AFOS			1				1	1		1	1	
Mobile reporting			5	25	25		5	5	15	5	5	5
Mobile direct			5	10	10		5	5	5	5	5	5
Coordinating				5						5		

Figure 5. - Effect of varying communications functions on spacecraft characteristics for reference spacecraft (three warning channels, five coordinating channels, 1543 watts of beginning-of-life power, cost of $27.8 million, weight of 1030.29 kilograms).

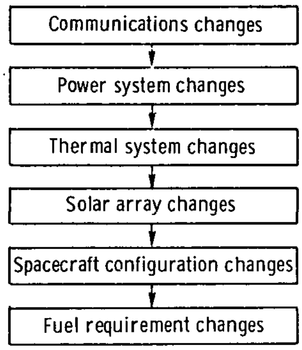


Figure 6. - Flow chart showing effect of changes in communications requirements.

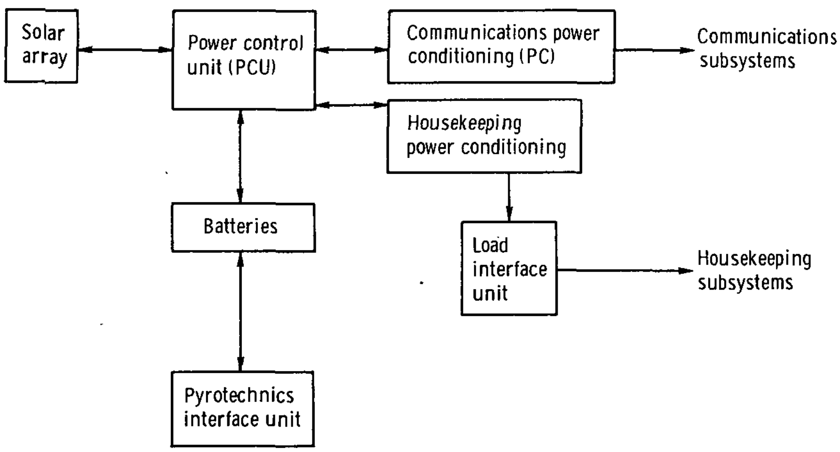


Figure 7. - Power subsystem configuration.

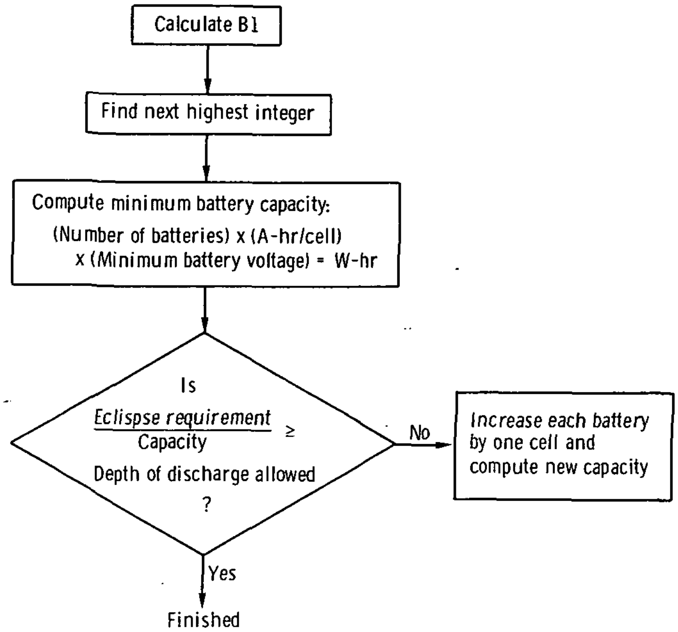


Figure 8. - Flow chart for battery sizing.

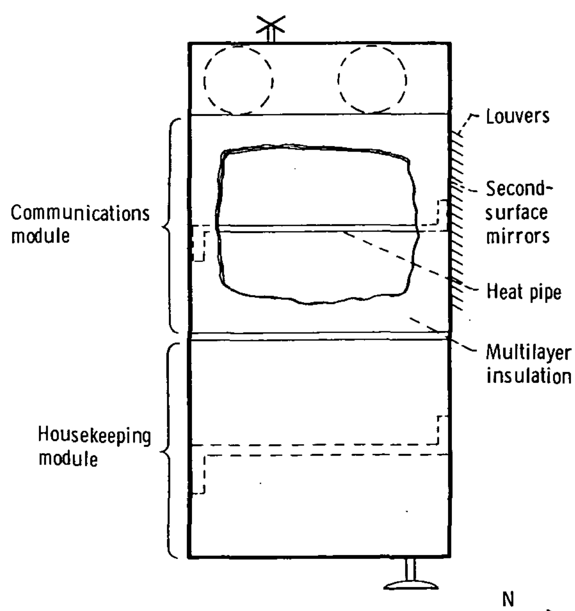


Figure 9. - Basic thermal system design.

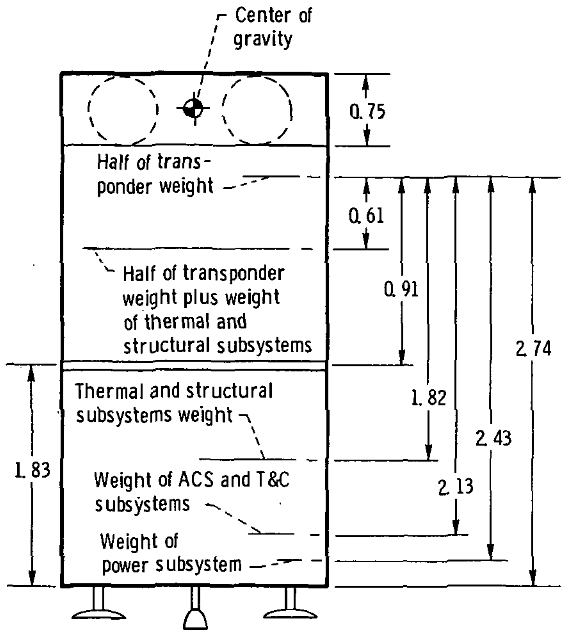


Figure 10. - Layout of Earth-viewing module. (Dimensions are in meters.)

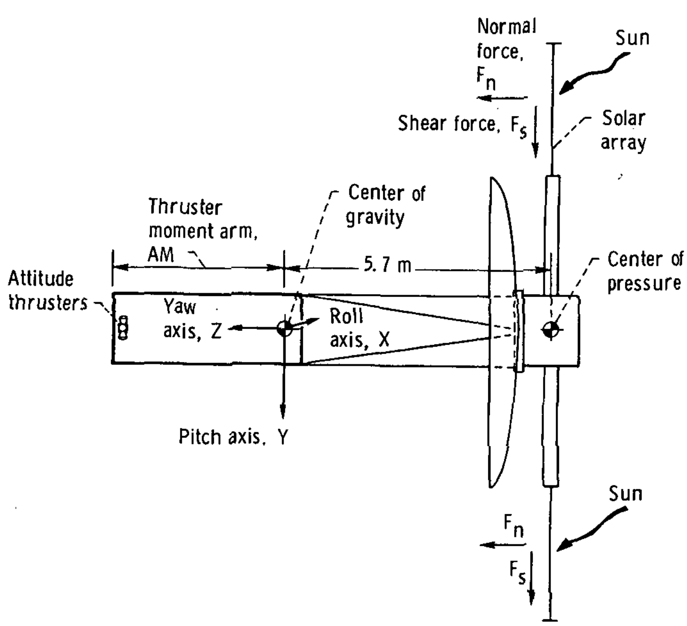


Figure 11. - Coordinate system, thruster location, and solar pressure forces, where X, Y, and Z denote roll, pitch, and yaw, respectively.

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